Bell 525 Flight Test Accident

Bell Helicopter Party Submission
DCA 16FA199

Bell Helicopter Textron Inc.
Bell 525 Serial Number 62001
N525TA

Italy, Texas
6 July 2016

Bell Helicopter
A Textron Company

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REFERENCES

[1] National Transportation Safety Board (NTSB), "NTSB No. DCA16FA199, 6 July 2016, Performance Study, Specialists Report," NTSB, Office of Research and Engineering, Washington, D. C..

[2] National Transportation Safety Board (NTSB), "NTSB No. DCA16FA199, 6 July 2016, Airworthiness Group Chairman's Factual Report," NTSB, Office of Aviation Safety, Washington, DC, 2017.

1. EXECUTIVE SUMMARY

On 6 July 2016, Bell Helicopter experienced the tragic loss of a prototype 525 helicopter, N525TA and its crew. The flight originated from the Arlington Municipal Airport (KGKY), as a developmental test flight and was conducted under the provisions of Title 14 Code of Federal Regulations Part 91. Visual Meteorological Conditions (VMC) prevailed at the time of the accident. In the months since the accident, a small team of Bell flight technology engineers, pilots, and flight test specialists have worked with NTSB investigators to determine the cause of the in-flight breakup near Italy, Texas.

The accident flight was a part of the company developmental test program to develop the 525 aircraft. The program consists of three development and envelope expansion (D&EE) aircraft. N525TA was aircraft number 1 in the test program. Due to the aircraft being a D&EE test vehicle, telemetry (TM) data streaming and on-board instrumentation and data capture devices provided a very large quantity of useful analysis data. Neither the cockpit voice recorder (CVR) nor TM cockpit voice channel were operative, so there is no voice or audio data from the test aircraft.

On the day of the accident, the twin engine helicopter was demonstrating the ability to recover from a single engine failure at a high airspeed, high power flight condition. Upon initiating the single engine simulation, the main rotor rotational speed (RPM) decelerated as expected. The crew lowered the collective control to reduce the power demanded by the rotor, initiating a recovery of the rotor RPM. The rotor RPM decay was arrested but not restored. While remaining at this low RPM, high airspeed condition, an unexpected vibration emerged.

The cabin vibration was amplified in a biomechanical feedback loop involving involuntary oscillations of the pilot collective control. Sensors utilized to stabilize the helicopter in gusts and maneuvers also participated in the feedback. This vibration grew rapidly as the aircraft continued to operate in a high airspeed, low rotor RPM condition outside the steady operating envelope of the aircraft. Approximately 21 seconds after the start of the test, amid a continued high vibratory environment, large control inputs coincided with the main rotor blades departing from their normal plane of motion, and the tailboom was severed. An inflight breakup of the aircraft ensued. Prior to the tailboom strike, the flight data indicates that the aircraft responded as expected to control inputs and there were no indications of structural or system failures.

The accident can be summarized as follows:
While at a high airspeed test point, a sustained low rotor RPM allowed for the development of high vibration throughout the aircraft. The high vibratory environment led to adverse control inputs that reduced the rotor RPM to critical levels and resulted in high flapping of the rotor blades. A near full aft cyclic control input led to one of the rotor blades severing the tail boom and loss of control of the aircraft.

Going forward, several changes to the aircraft are being implemented. In particular, the biomechanical and sensor feedback will be filtered by the control system so that these undesirable control inputs are not passed to the main rotor swashplate actuators, thereby preventing amplification of the vibrations present in this flight condition. The flight test program will assess filter effectiveness through a carefully planned approach as part of the completion of the remaining envelope expansion and certification testing.

2. ACCIDENT INVESTIGATION AND ANALYSIS

2.1 The Accident

At Bell Helicopter, test points are identified by the associated test (flight) number and the sequence in which test points are acquired, called the record number. The inflight breakup occurred during Test number 184 (Approximately the 184th flight of the test aircraft), and record 51 (Approximately the 51st test point planned for the day). The test point for record 51 was a simulated single engine failure test point at high forward speed. Upon initiating the simulated engine failure, main rotor RPM dropped to around 92% of nominal, and a vibration build-up occurred. The vibration led to control system feedback and excessive cabin vibration. As detailed in Section 3, "Accident Scenario Analysis," large control inputs resulted in significant main rotor blade flapping manifested when rotor rotational speed decayed below 80%. Approximately twenty-one seconds after record 51 began, a main rotor blade severed the tailboom from the aircraft.

The test aircraft's flight path for record 51 was approximately 320 degrees, according to GPS data. The test aircraft began the maneuver for Record 51 at approximately 3,000 – 3200 MSL. Record 51 was the last high speed test point for the given altitude. As the test aircraft nosed over to obtain the required airspeed, the chase aircraft positioned on the right side of the test aircraft. About 12-seconds into the test point, a "knock-it-off" call was transmitted over the radio by the TM room. An engineer at a TM monitoring station had observed high vibration in the test aircraft. The knock-it-off communication calls for the test pilot to restore the helicopter to a safe condition. The chase crew, consisting of a pilot and copilot, then observed the nose of the test aircraft come up slightly in an apparent attempt to "knock-it-off" and execute a deceleration. Additionally, the chase crew observed that the rotor system appeared to become "out of phase" with the airframe with one blade appearing to be flying significantly higher than the other four blades. The chase pilot then reported seeing the tailboom flex down and then back up, with the tailboom folding up and drifting away from the remainder of the aircraft. The chase aircraft then executed maneuvers to stay clear of the debris. As the test aircraft rotated, the aircraft began to break apart with the larger pieces continuing forward along the original flight path. Shortly after the test aircraft yawed left, the trajectory became steeper, with the test aircraft impacting the terrain left wing down.

Figure 1 shows the test aircraft's flight path for record 51. Each point is annotated with the time in seconds from the start of record 51 and with the corresponding height above ground in feet. The test record began at time zero, in the lower right of the figure. The telemetry system stopped recording data at approximately the 21 second mark. The test aircraft's last verified position is shown as "last point" in Figure 1 (from Reference [1]).

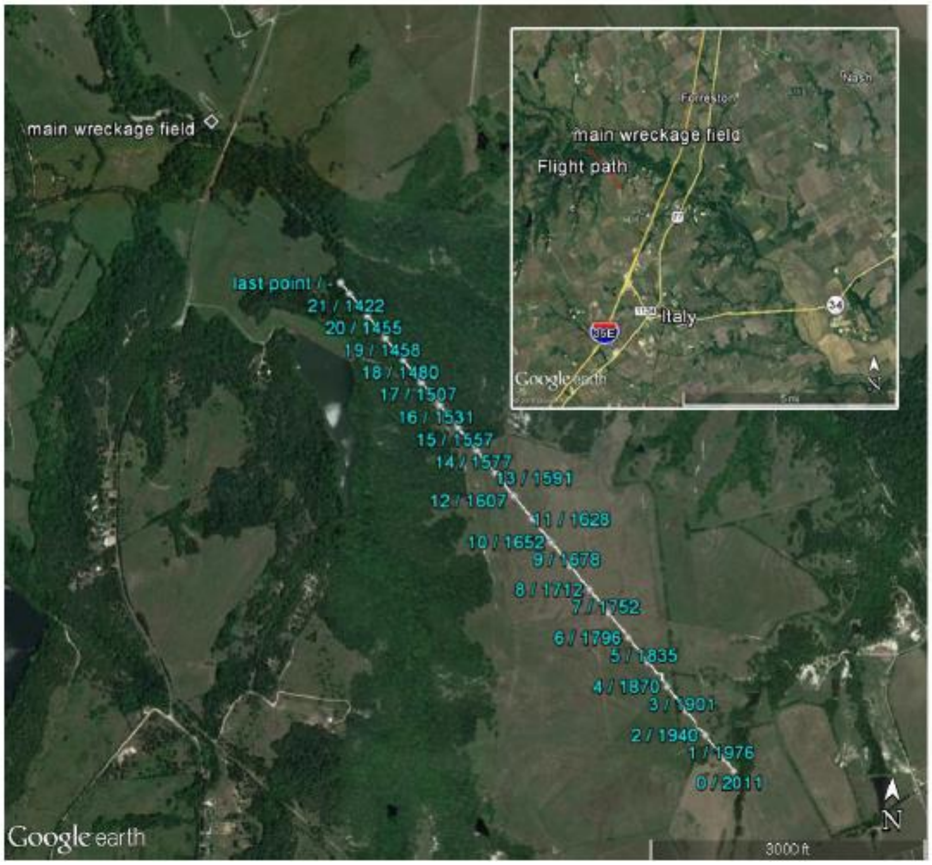

Figure 1. Test Aircraft Flight Path and Wreckage Location. (image courtesy NTSB)

The main wreckage field was 2,200 ft from the last transmitted GPS point, along the flight path heading of 320°. The wreckage was distributed into two distinct areas. The main wreckage site was comprised of an impact crater, remnants of the main fuselage, cockpit, transmission and main rotor hub, two of the five main rotor blades, the forward portion of the tailboom, and both engines. There was evidence of a post-crash fire at the main wreckage site. The wreckage debris path at the main wreckage site, about 200 feet in length, was oriented along the 315-degree magnetic bearing. The second distinct wreckage site, about 1,300 feet southeast of the main wreckage site, comprised the aft portion of the tailboom. The aft portion of the tailboom contained two of the tail rotor drive shafts, the intermediate gearbox (IGB), tail rotor gearbox (TRGB), and tail rotor hub assembly with all four blades and dampers attached.

Three of the five main rotor blades were found separate from the main and secondary wreckage sites. Various pieces of forward cowlings, cockpit frames, and cabin doors were found in a debris path between the main and secondary wreckage sites. Additionally, lightweight debris, such as insulation and main rotor blade skin pieces, were found scattered to the northeast of the debris path between the main and secondary wreckage sites, with the furthest piece being found about 1,520 feet away from the debris path between the main and secondary wreckage sites. A debris field extended between the last data point and the main wreckage, covering approximately 4,000 ft (north to south) by 1,700 ft (east to west) (see Figure 2, from Reference [2]). The tailboom, as well as the orange, white, and red main rotor blades were in the larger debris field while the root ends of the blue and green blades were with the main wreckage of the aircraft.

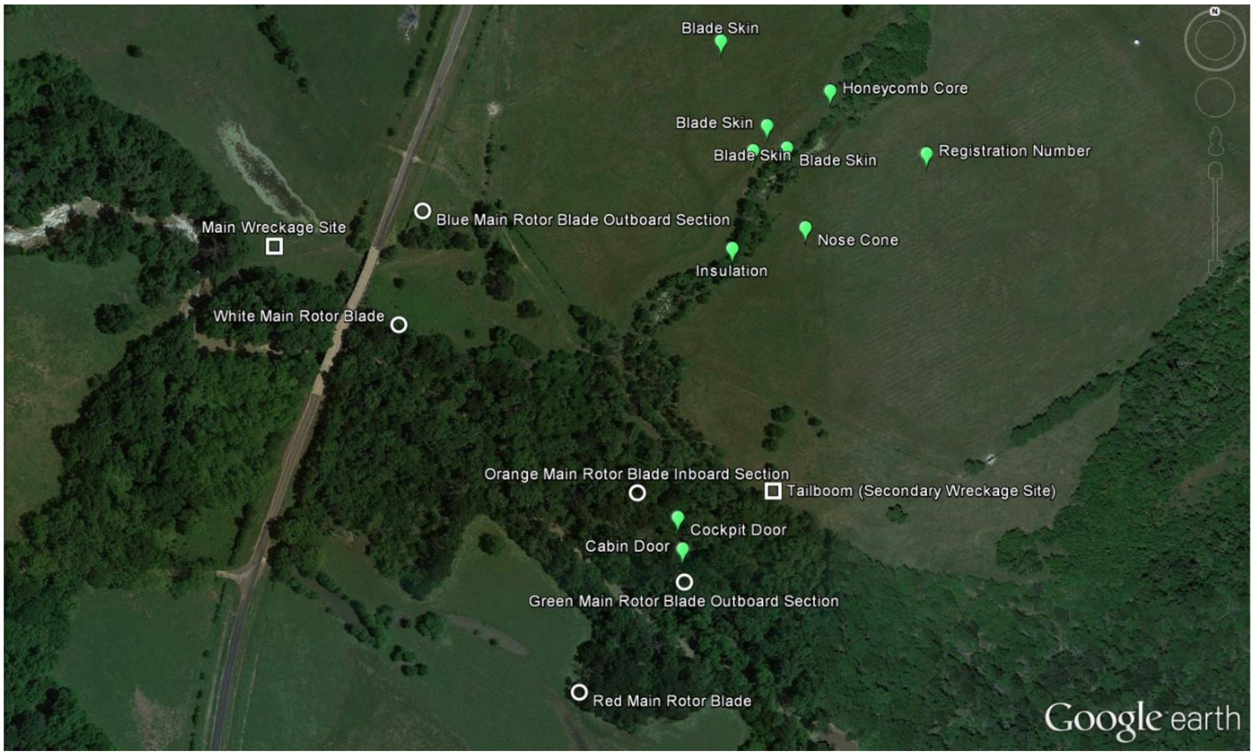

Figure 2. Accident Aircraft Debris Field. (image courtesy NTSB)

The wreckage was recovered and transported to a secure hangar at the Bell Helicopter Plant 6 located at Arlington Municipal Airport on 9 July 2016 for further examination.

2.2 Test Crew

2.2.1 Pilot in Command

The pilot, age 36, held an Airline Transport Pilot (rotorcraft) certificate, and a commercial pilot certificate with single-engine land and instrument airplane privileges, issued March 18, 2014, and a second-class medical certificate issued on April 8, 2016, with no limitations. Examination of the pilot's military logbooks, civilian logbook, and Bell Helicopter flight records revealed that he had 2,898.9 total flight hours, 2,744.6 rotary wing flight hours, 72.7 hours in the Bell 525, and

40.1 hours flown within the previous 30 days. He held a Letter of Authorization (LOA) from the Federal Aviation Administration dated December 2, 2015, authorizing him to act as pilot-in-command of the Bell Helicopter experimental helicopter designated model 525. His most recent flight review was conducted on October 29, 2015, flown in a twin engine Bell 430 helicopter. He completed Crew Resource Management (CRM) training January 12, 2015. The pilot received his Bachelor of Science degree in electrical engineering from the United States Naval Academy in 2002, and graduated from the United States Naval Test Pilot School (USNTPS) in 2010. After USNTPS he worked on numerous flight test projects involving the AH-1W and UH-1Y helicopters. On September 23, 2013 he joined the Flight Test Department of Bell Helicopter. His primary responsibility was being an experimental test pilot on the Bell 525 program.

2.2.2 Co-Pilot

The copilot, age 43, held an Airline Transport Pilot (rotorcraft) certificate, and a commercial certificate with single-engine land, multi-engine land, and instrument airplane ratings, issued October 15, 2014, and a second-class medical issued on May 31, 2015, with no limitations. Examination of the pilot's military logbooks, civilian logbooks and Bell Helicopter flight records revealed that he had 3,957.5 total flight hours, 2,589.4 rotary wing hours, 84.1 hours in the Bell 525, and 27.4 flight hours within the previous 30 days. He held a Letter of Authorization (LOA) from the Federal Aviation Administration dated December 2, 2015, authorizing him to act as pilot-in-command of the Bell Helicopter experimental helicopter designated model 525. His most recent flight review was conducted on November 6, 2015, flown in a Bell 407. He also completed a flight review in the twin engine Bell 430, October 6, 2015. Crew Resource Management (CRM) training was completed on January 12, 2015. The copilot received a Bachelor of Science degree from Texas Tech University in 1996, and completed US Navy flight training in 2000. In 2006 he graduated from the USNTPS and proceeded to work on numerous AH-1W and UH-1Y test programs. He joined the Bell Helicopter Flight Test Department on August 2, 2010. His primary responsibility was being an experimental test pilot on the Bell 525 program.

2.2.3 Telemetry (TM) Room

The TM room consists of multiple monitoring stations whose purpose is to provide real time observation of direct and derived parameters streamed from the aircraft to the TM room. During record 51, there were 9 stations being manned: The Test Director, Rotor Dynamics, Control Laws (CLAWs), Flight Control Systems, Structural Dynamics, Handling Qualities (HQ), Data Operations, Loads and the Telemetry Room Operator. These specialists are integral participants in ensuring the safe conduct of flight test. All TM personnel have the authority to stop a data point or an entire flight if a parameter exceeds a pre-determined value or if there is something out of the ordinary. The Test Director is the only one who has direct communication with the aircraft, but each member has the ability and duty to call a "knock-it-off" if either of these events were to occur.

2.2.4 The Chase Aircraft and Crew

The chase aircraft utilized on the day of the mishap was a Bell 429 helicopter. Two Bell pilots were in the aircraft. Their primary duties were to monitor the test area for other aircraft, monitor the flight for safety issues and observe the flight test vehicle as it executed the test card. The chase aircraft was in radio communication with both the test vehicle and the TM room. The PIC is a USNTPS graduate and experienced test pilot. The co-pilot is a Rotorcraft ATP and graduate of USN Post Graduate School Aviation Safety Course. At the higher speeds required for some test points, the 429 aircraft is unable to keep up with the test vehicle and is more of an area chase for those points. This is acceptable with proper in-flight planning and adjustments in space prior to commencing a test point at the higher speeds.

2.3 Environmental Conditions

The closest weather reporting location to the accident site was Mid-Way Regional Airport (KJWY), located approximately 12 miles north of the accident. The airport had an Automated Weather Observation System (AWOS). The following conditions were reported near the time of the accident: 1155 CDT, automated, wind from 180° at 15 knots gusting to 20 knots, visibility 10 statute miles, scattered clouds at 3,000 feet AGL, temperature 32° C, dew point temperature 24° C, altimeter 29.97 inches of mercury (Hg). The observation provided a calculated relative humidity of 63%, a station pressure of 29.19 inches Hg, with a density altitude of 3,175 feet. No precipitation was recorded at the station.

Additional weather reporting in the general vicinity was from Hillsboro Municipal Airport (KINJ) located 15 miles SSW of the accident site. At 1156 CDT METARS reported temperature as 31°C (87°F) with a dew point of 23°C (73°F) and an altimeter setting of 29.98. Visibility 10 statute miles with scattered clouds at 3,000 ft. Winds were reported at 14 knots from 170°, gusting to 22 knots.

Based on interviews with the chase pilots and the TM crew there had been discussion about the air quality during the conduct of the test. The team had short discussions on the matter and decided that the air quality was sufficient to continue with the test flight. The accident occurred on a second attempt at this data point as the first attempt was discontinued due to the test aircraft encountering a thermal or choppy/turbulent conditions. Environmental conditions of the day are not deemed to be contributory to the accident.

2.4 The 525 Program

The Bell 525 program is a type certification program of a new type design helicopter. The aircraft is designed as a transport category helicopter for the super medium twin helicopter market. The originally planned test program consisted of 5 test vehicles; 3 dedicated D&EE

aircraft and 2 dedicated production representative aircraft. The accident aircraft was 525 number 1 and was one of the D&EE aircraft. At the time of the accident, the program had flown the three D&EE test aircraft and had accumulated a total of approximately 300 hours of flight time with over 140 hours of ground test activity.

2.4.1 The 525 Aircraft

The Model 525 aircraft features a five bladed main rotor, canted four bladed tail rotor, composite airframe, a triple redundant Fly-By-Wire (FBW) Flight Control System (FCS), a Garmin G5000H Integrated Flight Deck, a Honeywell RE100BR Auxiliary Power Unit (APU) and two General Electric (GE) CT7-2F1 engines. Additionally, the aircraft was equipped with retractable tricycle landing gear. The aircraft is designed to carry 16 passengers and a crew of two at a cruise speed of 155 knots and operate up to an altitude of 20,000 ft at a maximum takeoff weight of 20,500 lbs.

The 525 Integrated Flight Deck is configured with two pilot stations and a console between the seats. Each pilot station has a cyclic side stick controller forward of the seat's right armrest and a collective side stick controller immediately forward of the seat's left arm rest. Each pilot has a set of directional (anti-torque) pedals forward of the seat. The instrument panel consists of four identical primary flight displays. The center console consists of two Garmin Touch Control (GTC) panels, landing gear handle, and Nav/Com panel. The engine controls, known as the Crank, Off, Start, Idle, Fly (COSIF) knobs are directly above the GTC's. There is a single COSIF knob for each engine.

2.4.2 The Test Aircraft

The test aircraft, serial number (S/N) 62001 was manufactured in 2015 and was the first 525 test vehicle. It had flown approximately 200 hours prior to the accident flight. As the primary D&EE air vehicle, the test aircraft was instrumented to capture 3000 measurements and 5200 parameters, including derived readings. Each parameter was sampled and recorded by the data system between 31 and 1000 times per second (Hz). These data were both recorded and stored on the onboard flight test recorder system, as well as transmitted via TM to the ground station at the Bell flight test facility.

At the time of the accident, the test aircraft was configured for a heavy gross weight (approximately 19,975 pounds) and a forward center of gravity (CG). In addition to the standard Flight Deck configuration, the test aircraft was configured with a Flight Test Switch Panel located on the center console. Among other items, this panel included controls for the One-Engine-Inoperative (OEI) special training mode. Additionally, outboard of the instrument panel, each pilot had a pilot display unit (PDU) that provided a limited number of real time flight test instrumentation parameters.

2.5 Flight Test Processes and Preparation

2.5.1 Bell Flight Test

Bell has a robust D&EE test process that includes not only test and evaluation at the aircraft level, but also test and evaluation of the individual components and systems. The use of analytical and physical tools to test and analyze components, systems, and systems-of-systems are prerequisite to air vehicle ground and flight test. Risk analysis and management is initiated early in the process and iterated upon as the vehicle and systems change or as more is learned from test and evaluation activities. Risk analysis and management is an integral component of the development and flight test process.

2.5.1.1 Systems Integration Lab (SIL)

Bell Helicopter typically utilizes an engineering SIL to evaluate both software and certain hardware prior to items being introduced to flight test. The specific SIL for the 525 program is the Relentless Advanced Systems Integration Lab (RASIL). The RASIL consists of an accurate engineering representation of the cockpit, including control feel and visual in-flight representation projected on a wrapped screen. Next to the RASIL cockpit is a room containing actual flight hardware rigged to apply flight loads into engineering representations of related hardware. Control movements in the cockpit cause the corresponding hardware to respond. The RASIL is used not only to validate software but also as a training and risk mitigation tool as it allows the pilots to develop flight procedures and pre-fly flight test cards prior to test execution in the air vehicle.

2.5.1.2 Flight Simulation

Bell utilizes an engineering flight test simulator to assist in the development of cockpit ergonomics, layout, and control laws. The Bell 525 program utilized the engineering flight simulator early in the development of the program to assist in designing and evaluating the overall cockpit layout as well as the initial control law (CLAW) development. These development activities transitioned to the RASIL once the RASIL was functional.

2.5.1.3 Flight Test Risk Management

Bell Helicopter has a robust Flight Test Risk Management Plan (FTRMP) that is in alignment with FAA Order 4040.26. The flight test risk management process starts early in a program's life cycle and continues until the program is complete. Flight Test Risk Worksheets (FTRW) are completed for each identified air vehicle test risk. Each identified risk is evaluated and run through a mitigation process. The final version of the FTRW is approved at the appropriate level, based on the initial risk category. Flight test risks are briefed during the program review process at the appropriate readiness review for a flight test stage and also at each individual flight pre-flight briefing where that risk is applicable. Flight test risks are cataloged by program so that they may be utilized for reference and assist in the identification and mitigation of risk on future programs.

2.5.1.4 Safety Management System

At the time of the accident Bell was in the process of developing a Safety Management System as part of the Federal Aviation Administration's (FAA) Design and Manufacturing Safety Management System (D&M SMS) Pilot Project. At that time, the plan was to implement an Aviation Safety Management System (AVSMS) under that umbrella for future acceptance by the FAA. The tools of an SMS undergoing implementation at the time of the accident were the Just Culture System, the Confidential Reporting System and the Flight Risk Assessment Tool (FRAT). The Just Culture Process is not germane to this accident and the Confidential Reporting Tool was not implemented at the time so neither will be discussed further in this report.

At the time of the accident Bell was in the final stages of implementing a FRAT as part of our AVSMS. In reviewing the relevant data, there appears to be nothing that would have identified an elevated risk on this day.

2.5.2 Day of Activities

2.5.2.1 Flight Briefing

Test flight 184 was briefed by the test team at the flight test facility in Arlington, TX at 0600 on the date of the accident. The purpose of the test was D&EE, specifically: Heavy forward CG engine load and vibrations at maximum continuous power, heavy forward CG longitudinal roll spot check, 2 engine to 1 engine simulated engine failures, and run-on landings up to 60 knots. The test was intended to begin at 0730 but was delayed due to weather concerns. At approximately 1000, based on the improved weather conditions, the team decided to conduct the test flight.

2.5.2.2 Flight Test Risks

Two specific flight test risks were identified for test flight 184 (Table 1).

Table 1. Flight Test Risks Identified for Test Flight 184

Hazard	Mitigation
Rotor speed droops below safe rotor controllability speed	• Adherence to best practices and approved test procedures • Autorotation practice conducted in the RASIL • Entry conditions will begin in the middle of the weight and airspeed envelope; build up in GW, airspeed and entry power levels separately • TM monitoring • Buildup in delay time before pilot response • Execute power recovery procedures before main rotor RPM is allowed to droop excessively
Transmission and engine overtorque/overtemp and/or overspeed due to sudden increase in power available	• Aircrew familiar with Single/Dual Engine Emergency Procedures • Adherence to best practices and approved test procedures • Evaluation of software in RASIL prior to test • Conduct restrained ground run tests before flight to verify software functionality • Conduct dual to single points with engine at idle prior to engine off points • Buildup in airspeed and entry power levels separately • Verify GE Trim file FADEC channel in brief • Dual concurrence before any channel, OEI training selections

3. ACCIDENT SCENARIO ANALYSIS

The accident occurred during a developmental test flight for evaluating recoveries from simulated single engine failures. In testing, single engine failure is simulated by a Special One Engine Inoperative (OEI) training mode that permits both engines to run at reduced power levels, matching the transient and sustained power loss associated with a single engine failure. Engagement of the OEI training mode is via a touch control panel accessed by the pilot or co-pilot. The developmental testing on July 6, 2016 was conducted to examine recovery from simulated single engine failure for the heavy, forward cg, high airspeed portion of the 525 flight envelope. Below approximately 130 knots, the aircraft is capable of continued flight on one engine. In the higher speed portion of the flight envelope, power from only a single engine is insufficient to maintain the flight condition, and consequently the pilot must develop descent rate and/or reduce speed to initiate recovery. Since engine power for flight is delivered to the main rotor, a reduction in power available will result in a reduction of rotor rotational speed (RPM). The pilot manages rotor RPM decay by reducing the main rotor collective control. The main rotor collective pitch reduction initiates (or increases) the descent rate of the helicopter and immediately reduces the rotor power required. When the power required is less than the power available, the rotor RPM can be restored to the full value. For developmental testing on the day of the accident, the test point was considered complete once the rotor rotational speed achieved the target value of 103% RPM.

Simulated OEI testing at various altitudes and airspeeds had already been completed for two cases of aircraft loading (gross weight and center of gravity) as seen by the green points in . No anomalies were seen. The test aircraft was configured for heavy, forward cg loading on the day of the accident. Simulated single engine failure testing began at 4000 feet altitude for increasing airspeeds (10 knot intervals, beginning at 155 knots). Testing progressed to the final airspeed point for the given altitude when the accident occurred. The accident point is shown as the red dot in Figure 3.

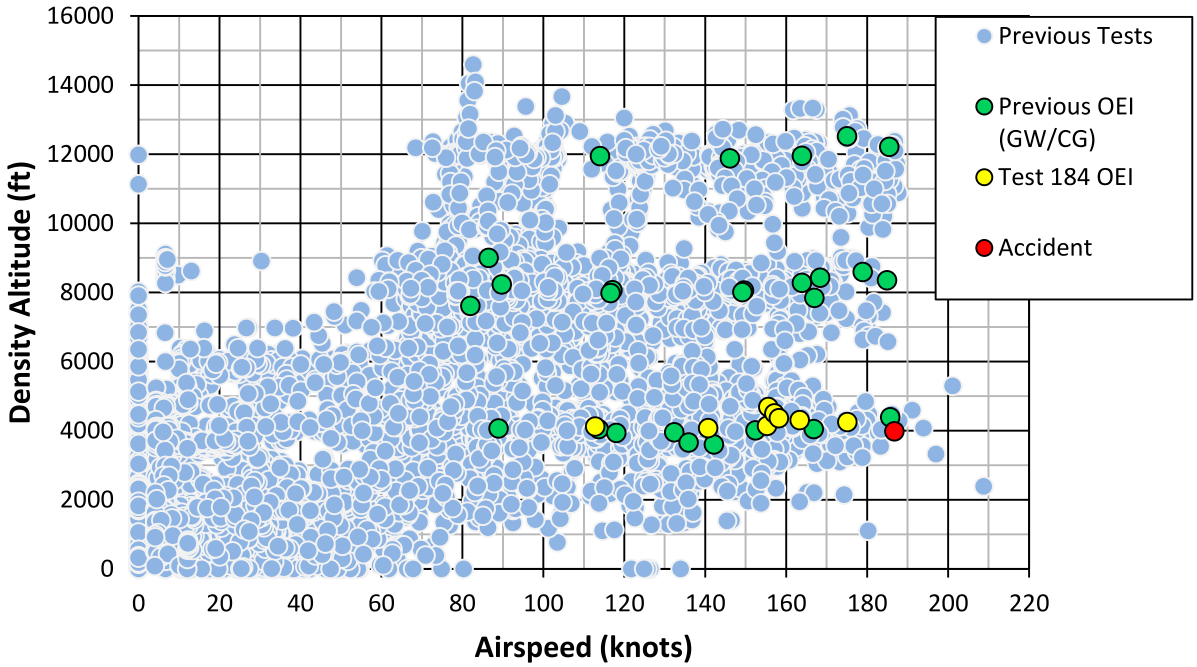

Figure 3. Envelope expansion points completed up to Day of Accident included numerous simulated single engine failure test points (in green). The accident occurred on Test Flight 184, which included similar simulated single engine failure test points (yellow) and the accident test point, record 51 (red dot).

For each airspeed, including the accident point, a test record[1] was initiated from steady flight conditions. The initial airspeed for the accident record was 185 knots and required a slight descent rate. Main rotor RPM, collective control position, fore/aft cyclic, and pilot seat vibration for the accident record are plotted in Figure 4. Referring to the upper sub plot of main rotor RPM, the OEI training mode was engaged by the co-pilot a few seconds into the record. Per test protocol, the pilot held the controls fixed for a 1-second "pilot delay." During the delay period, the engine power transitioned to the single engine power level. Since the accident record was initiated at an airspeed that required both engines, the rotor RPM decayed as anticipated. After the delay period, and referring to the second sub plot of Figure 4, the pilot reduced the collective control, arresting the decay of rotor RPM at about 91% of nominal rotor (RPM). The rotor RPM settled at approximately 93% RPM for about 6 seconds. Previous simulated single engine failure test records, shown in Figure 5, are typical and indicate that a second reduction (or continued gradual reduction) in the rotor collective control may be necessary to completely recover rotor

[1] At Bell Helicopter, flight data are recorded continuously for the entire test flight, but key maneuvers (test points) are separated into events by a "record number" that is indexed automatically each time the pilot depresses a specific button on the control stick. The record identifier number is included with the recorded flight data (typically until the maneuver is complete) and the button is pressed a second time. Record numbers allow test points to be easily retrieved by flight analysts, without resorting to tedious identification of particular time slices in the recorded data.

RPM to the target value of 103%. The modulation of the collective control allows the pilot to assess the rotor RPM response incrementally, whereas a larger control input may lead to unnecessarily larger excursions in rotor RPM. Referring back to **Figure 4**, the rotor RPM never recovers higher than 93% RPM for the remainder of the accident test record. The extended time at low rotor RPM represented an unanticipated steady operation outside the normal flight envelope.

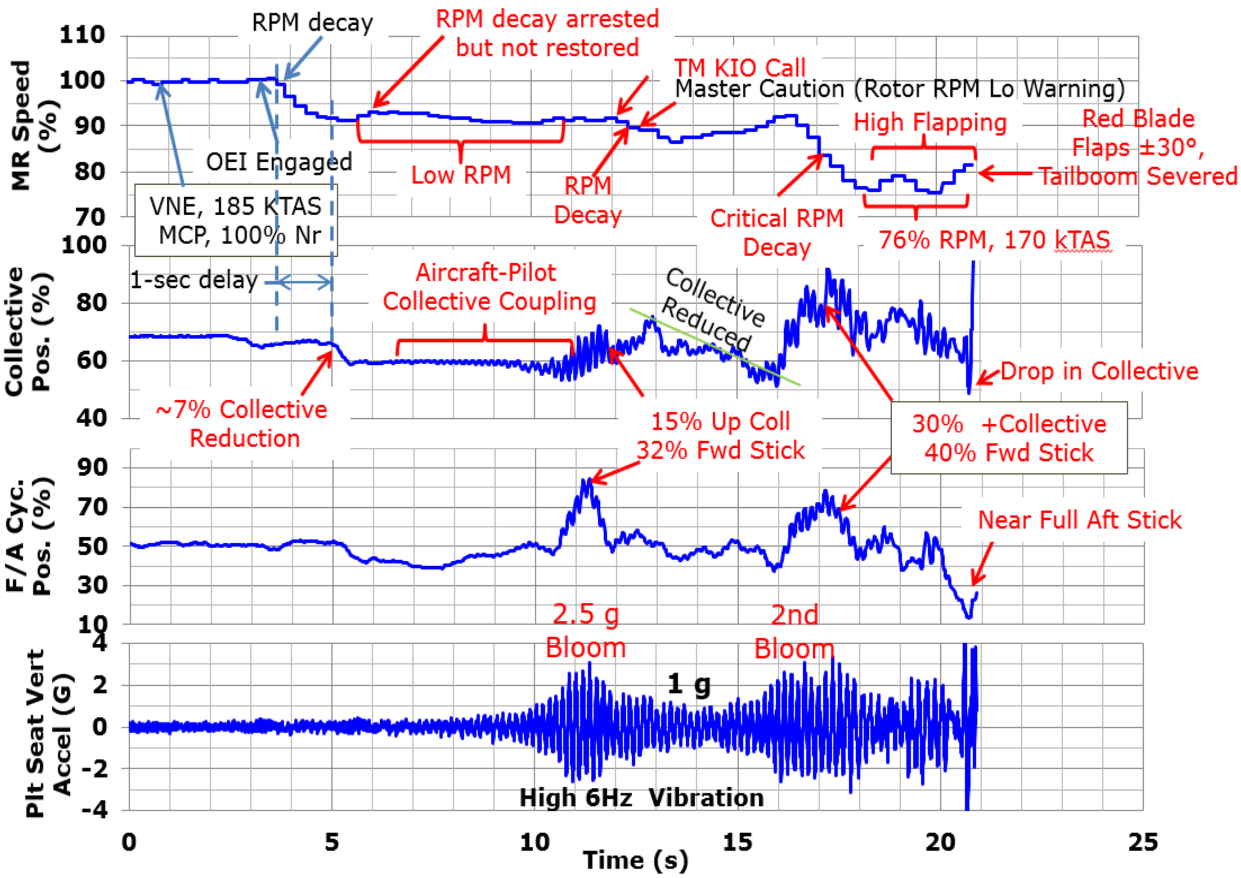

Figure 4. Time history of accident record, showing control inputs, rotor response, and pilot seat vibration level.

Still referring to **Figure 4**, while in the region labeled "Low RPM," a further reduction in collective control might have been anticipated (to fully restore rotor RPM)[2]. As discussed later, an in-depth analysis of the flight record shows that the sustained low rotor RPM was linked to the development of a main rotor dynamic mode and a feed-back loop that produced high vibration levels throughout the aircraft. Referring to the bottom plot in **Figure 4**, the vibration level sensed by the pilot seat accelerometer began to show vibration rising above the background

[2] The pilot may have held controls fixed to assess the emerging vibration before making further changes to complete the maneuver.

level at about 6.5 seconds. In Figure 6, a 6 Hz band pass filter is applied to the vertical vibration signal to highlight the predominant 6 Hz frequency of the vibration. In referring to the second subplot of **Figure 4** (labeled "Aircraft-Pilot Collective Coupling), the vertical vibration of the pilot seat generated an involuntary 6 Hz biomechanical oscillatory motion of the collective control position that further amplified the vibration level and (with other factors discussed later) resulted in the first vibratory bloom (labeled "2.5*g* bloom") in the lower subplot of **Figure 4**. The 2.5*g* vibration bloom (at 11 seconds and again at ~17 seconds) was nearly 40 times the normal vibration level seen in the experimental aircraft.

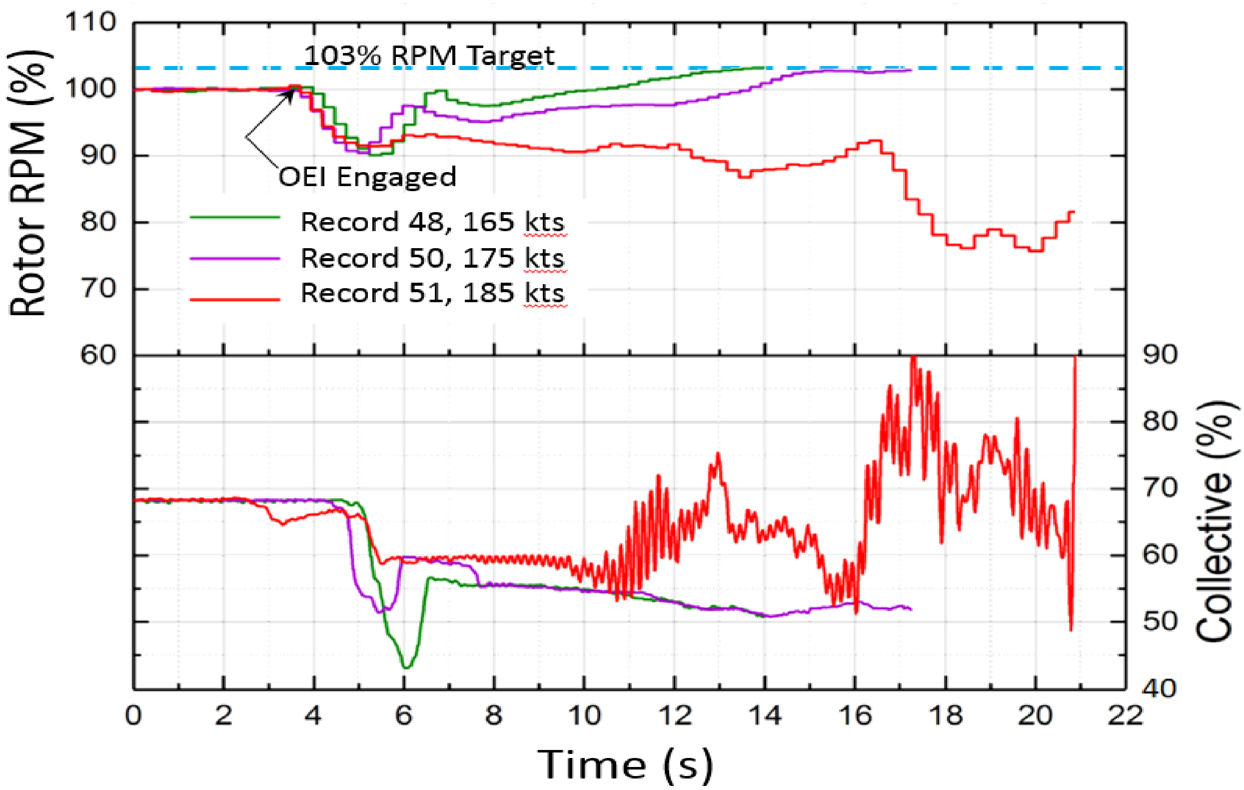

Figure 5. **Completed test records 48 and 50, just prior to the accident record (51), showing rotor RPM recovery to 103% RPM with modulation of collective control.**

Referring to the collective control position in **Figure 4**, it is seen that the collective (while undergoing 6 Hz involuntary oscillation) also undergoes an increase in mean position (from 10.5 seconds to 11.5 seconds). It is believed that the mean increase in collective position is related to the pilot undergoing significantly increased vertical vibration, and the pilot may have been unaware of the command. Nevertheless, the increased collective position represents an adverse increased power requirement by the rotor. With only single engine power available, a reduction in RPM is evident in the main rotor (see the top of **Figure 4**).

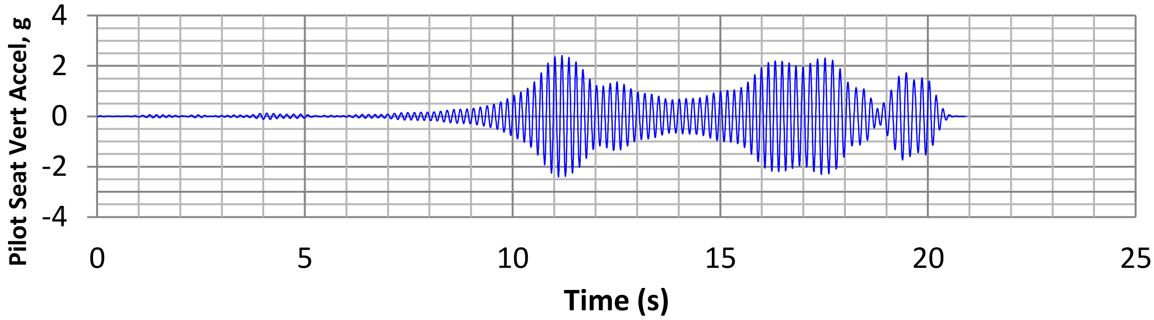

Figure 6. Pilot seat vertical vibration with band pass filter showing 6 Hz signal content.

While the mean collective increase is likely involuntary, the forward cyclic applied between 10.5-12 seconds appears to have been a deliberate input. While counterintuitive to the recovery, the forward cyclic input may have been an attempt to bring the aircraft to a nose down pitch attitude that existed earlier in the maneuver when the vibration was low[3]. Pitch attitude, shown in Figure 7, indicates that the maneuver was entered in a slight dive (~9 degrees nose down just prior to the 4-second mark). There was no particularly noticeable vibration in the dive, but at the 10 second mark, when the vibration had become excessive, the pitch attitude was approximately 2-degrees above the horizon. As seen in **Figure 4**, the vibration levels were reduced noticeably (but were still high) after the forward control input, but for different reasons that will be discussed later.

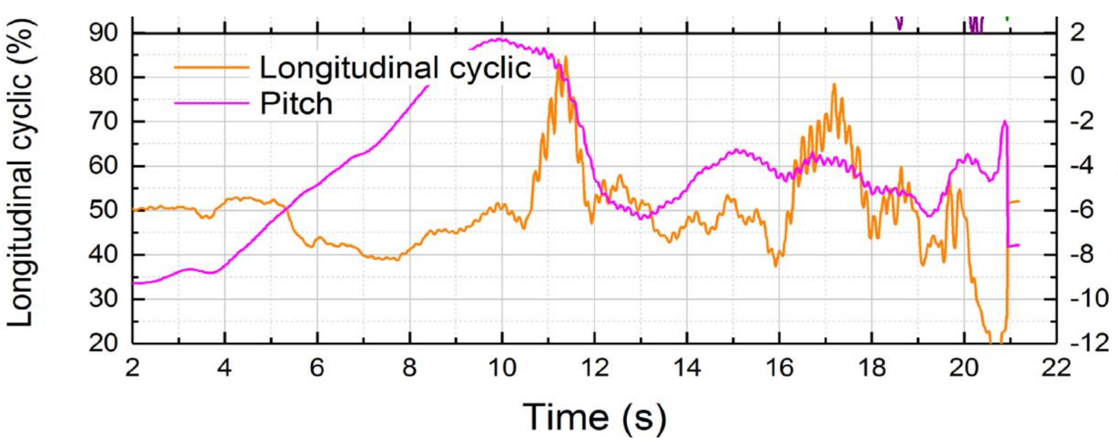

Figure 7. Aircraft pitch attitude change (right Axis) coincided with increased vibration and may have led the pilot to apply forward cyclic during the accident record.

[3] A single engine failure encountered at high forward speed would typically be followed by gradual aft cyclic control to slow down the aircraft and achieve an airspeed where flight can be sustained on one engine.

The first vibration bloom (at ~11 seconds) shook the entire aircraft, creating multiple warnings and alerts for the telemetry ground crew. A "knock-it-off" call was issued to the aircrew at approximately 12 seconds. In general, a "knock-it-off" call given during a maneuver calls for the pilot to restore the helicopter to a safe condition. Vibration levels initially came down, but remained high (at 1*g*) after the call.

Because the collective control position had increased, the main rotor RPM dropped below 90% at about the 12.5 second mark. The main rotor RPM drop triggered a Master Caution on the pilot's CAS display with a "Rotor RPM LO" warning. The message would have coincided with a general audio "bong" due to the low rotor RPM. As seen in **Figure 4**, the pilot lowered the collective control for the next three seconds in an apparent response to the Master Caution. The rotor responded with an increase in RPM (toward the 92% level at 16 seconds), which coincided with another increase in the vibration level. The second bloom in vibration (at 16 seconds) appears to have led the pilot to change the direction of the collective control from reducing the collective (and increasing RPM) to increasing the collective. The increase in collective at this time caused a critical decay in rotor RPM that resulted in a rotor rotational speed near 76%, with the aircraft still above 170 knots.

In this flight condition, the main rotor blade coning angle increased and the blades developed high out-of-plane flapping motion. The low rotor RPM, collective feedback, and rapid control inputs resulted in blades flapping independently and out of the normal rotor disk plane as discussed later. A near full aft cyclic control input, (**Figure 4**, at the end of the record) resulted in a single blade striking and severing the tailboom. The chase aircraft reported seeing the rotor "wobble" and a blade "flying out of track." The flight data recording and telemetry ceased almost immediately after the tailboom strike, with the aircraft still above 170 knots and 1400 ft above terrain. The chase pilot maneuvered to avoid the debris field as the tailboom departed. Without its tail the test aircraft was no longer controllable. Multiple blade strikes on the forward cabin were evident from the wreckage, but were not part of the recorded data. The tailboom landed largely intact, some distance from the main wreckage, with a ~30 degree cut-line near the root of the boom and no evidence of multiple blade strikes. The main transmission, rotor hub, and some blade root sections, still attached to the hub, were at the main impact crater with the cabin. The engines were also near the main impact site. The telemetry stream and the data recording indicated that there were no structural failures, no electrical system failures, and no hydraulic system failures until the last fraction of a second when the aft cyclic control input led to severing the tailboom. The control system continued to provide direct pilot control of the main rotor and tail rotor until this time.

While it is understood that the combination of low (76%) rotor RPM at high forward speed and a large aft control input was the immediate cause of the in-flight breakup, the underlying high amplitude, 6 Hz vibration and oscillatory control feedback led to the flight envelope departure. The previous 300 hours of envelope expansion testing had cleared the flight envelope to 12,000 ft altitude and 185 knots for combinations of loading. Envelope expansion involves extensive rotor dynamic assessments, where the rotor controls are dynamically stimulated and the system damping is confirmed. Testing showed that the rotor modes were well damped at 100% RPM, throughout the flight envelope. Although a period of low rotor RPM can occur for this type of maneuver, the sustained departure from nominal rotor RPM to 92% RPM while at high forward speed is a key aspect of the accident. As discussed earlier, the simulated single engine failure test point is considered complete when (after the power loss) the collective control has restored the rotor to nominal RPM. Upon power loss, the rotor RPM generally follows a deceleration/acceleration pattern that is considered transient while the recovery is underway. For the accident record, the decelerating transient is apparent (and is consistent with the collective control input), but the accelerating phase never increased the rotor speed above 93%. The resulting ‘steady state’, low RPM condition, initiated a sequence of events that amplified cabin vibration by coupling a main rotor dynamic mode with the airframe via control system feedback.

The behavior of the main rotor is governed by various dynamic modes. As a rotor turns, the blades participate in various patterns of lead-lag motion (modes). The frequency of each mode is primarily a function of rotor RPM, blade mass/stiffness properties, and lead-lag damper properties. While the mode frequencies can be predicted, the amplitude of motion is a more complex phenomenon, driven primarily by a rotor blade’s periodic aerodynamic excitation force. Mode responses are evaluated in flight testing to show that sufficient damping exists over the operating envelope. Each rotor mode can be expressed in terms of a rotating reference frame (attached to the rotor hub), or the airframe fixed (non-rotating) reference frame. The fixed frame of reference is used below to facilitate a description of the events in the accident record.

Frequencies of the main rotor and tail rotor modes of interest are shown in terms of the non-rotating (fuselage fixed) reference frame in Figure 8. The airframe first vertical bending mode frequency is also shown by the dashed horizontal line. Since fixed system rotor mode frequencies are a function of rotor RPM, the rotor modes have a slope, whereas the airframe represents a non-rotating structure, unaffected by rotation, and does not change with RPM. Figure 8 does not assign amplitude to any of the modes and only shows the modal frequencies. Referring first to the tail rotor, the accident record indicates that the tail rotor cyclic regressing mode was the predominant tail rotor “blade pattern” as the rotor decelerated to 93% RPM. At the same time, the main rotor’s cyclic regressing mode (near 2.4 Hz) was the dominant pattern[4]. The airframe first vertical bending mode frequency (verified by airframe vibration tests) was near 5.4 Hz.

[4] Multiple rotor modes are present at any given time. The dominant mode is determined from measured damper motion.

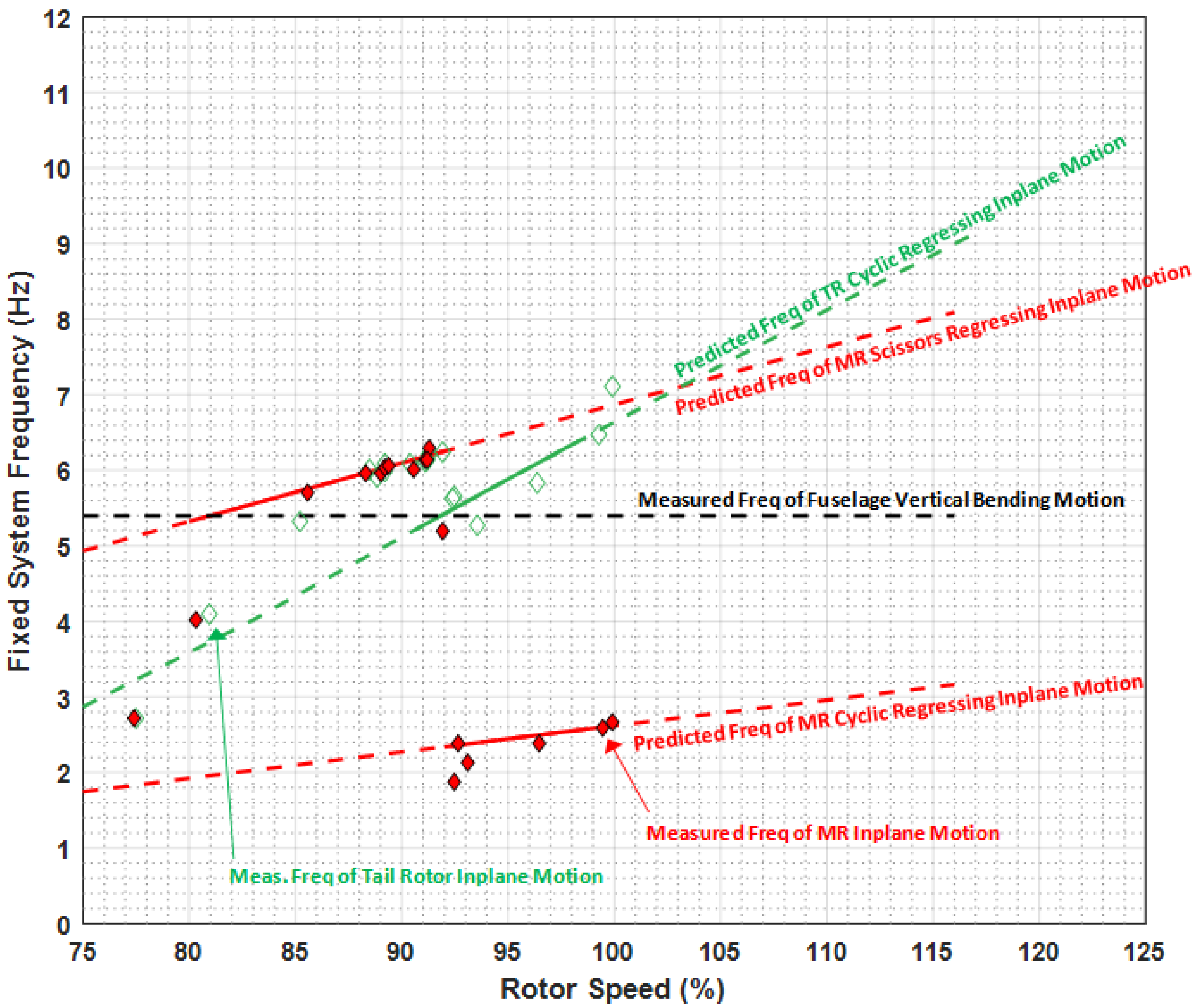

Figure 8. Airframe, Main Rotor, and Tail Rotor dynamic modes predicted and measured during the accident record.

Figure 4 showed that after engaging the OEI training mode and arresting the rotor RPM decay, the rotor rotational speed remained below 93% RPM for the remainder of the record. The continued low rotational speed of the rotor led to high vibration levels throughout the helicopter. In the fixed frequency chart (Figure 8), it is seen that at this low RPM, the tail rotor cyclic regressing mode frequency crosses the 5.4 Hz first vertical bending mode frequency of the airframe. Of itself, the crossing of frequencies is not an issue since the behavior of the tail rotor and airframe system is well damped, with vibration levels at the 5.4 Hz frequency quite low (see Figure 6 prior to 6.5 seconds). At about 6.5 seconds, the dominant main rotor phasing transitions to the scissors regressing mode. With the scissors mode dominant, all measured frequencies shift to the ~6 Hz scissors mode frequency. The scissors mode produces fore/aft oscillatory loading of the mast through aerodynamic forces.

The vertical bending mode of the airframe includes fore/aft pitching motion of the main rotor mast and vertical/lateral motion of the forward cabin. The motions of the airframe are illustrated in Figure 9, which shows (in highly exaggerated displacement) an overlay of the airframe at its minimum and maximum displacement for the first vertical bending mode. The forward part of the cabin (which includes the pilot seat and the attitude heading reference systems) sees mainly 6 Hz up-down vertical motion. In contrast, the response of the main rotor mast to fuselage bending is primarily fore/aft pitching. With reference to the main rotor tip path plane (not shown), the mast's fore/aft pitching motion is equivalent to the application of fore/aft cyclic. At high forward speed, cyclic control inputs of this nature constitute a stimulus for the main rotor scissors mode.

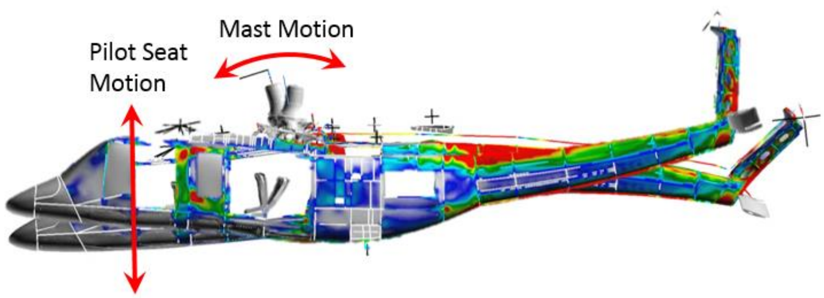

Figure 9. Illustration of the airframe first vertical bending mode (with highly exaggerated displacement) shows the forward cabin motion as primarily up and down, while the main rotor mast pitches fore/aft.

To further describe the progression of the accident record, the pilot seat vertical vibration amplitude is added to the fixed system frequency chart of Figure 8 in Figure 10. The pilot seat vertical vibration level is denoted by the scale to the right of the figure, while the timing of the event is denoted by 1-second interval markers that coincide with the time axis of **Figure 4**. It is seen that vibration levels initially remain low, even after the main rotor transitions to the scissors mode. However, Figure 10 shows that with the main rotor dominated by scissors response, the character of vibration for the pilot seat begins to respond at the scissors mode frequency.

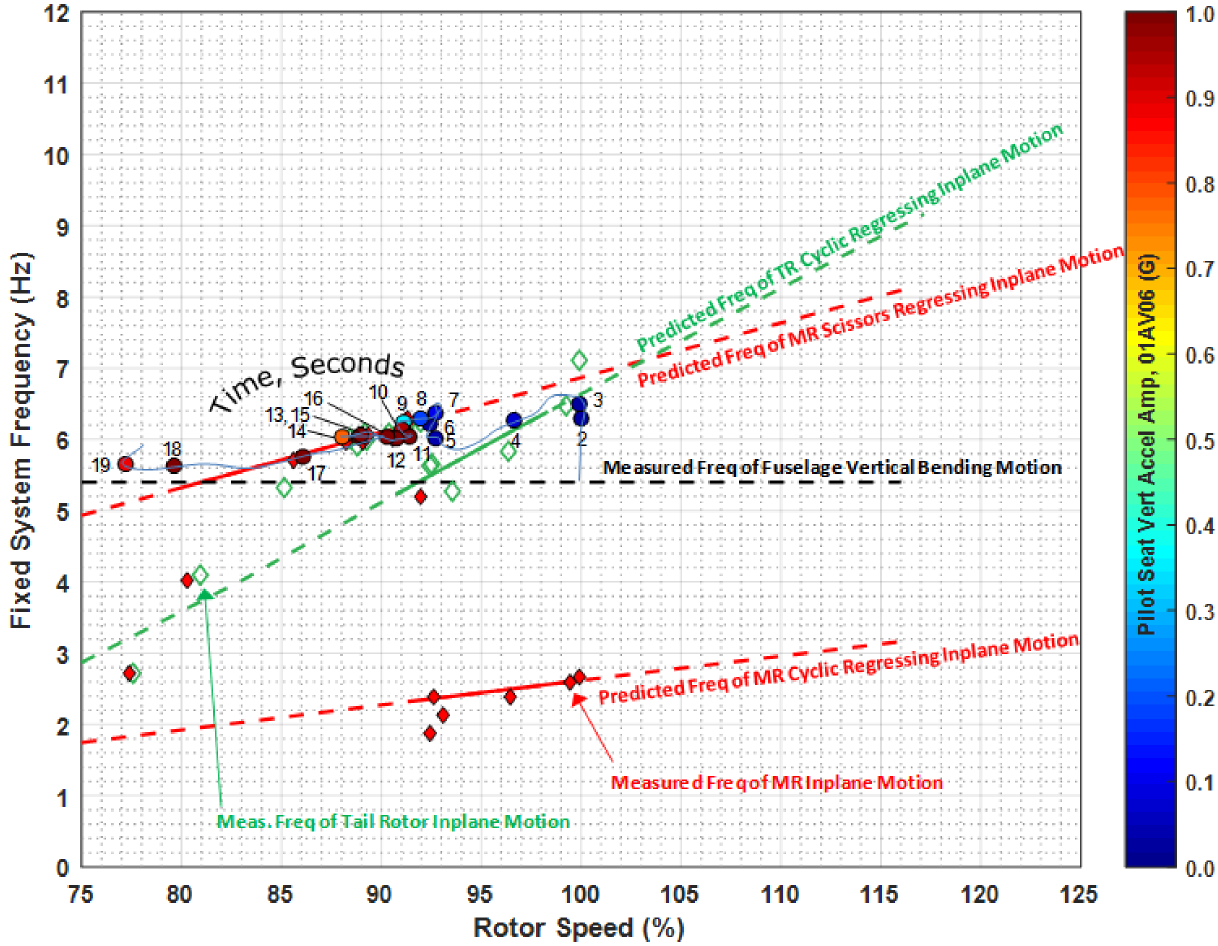

Figure 10. Modal frequencies of the Airframe, Main Rotor, and Tail Rotor with amplitude of Pilot seat vertical vibration.

Figure 10 shows that the ~6 Hz main rotor scissors mode is the predominant rotor pattern at approximately 6.5 seconds into the accident record. Pilot seat vibration is low at 6.5 seconds but the scissors mode feeds a low amplitude ~6 Hz vibration through the airframe bending motions. Aerodynamics of the scissors mode produces a fore/aft forcing of the main rotor hub at the scissors mode frequency. Fore/aft motion of the hub, in turn, transfers mast pylon rocking motion to the airframe resulting in 6 Hz vertical motion at the pilot seat and at the location of the Attitude Heading Reference System (AHRS) in the forward cabin. Airframe vertical bending couples with airframe lateral bending and some lateral 6 Hz motion is also produced in the airframe.

As shown in Figure 10, the 6 Hz pilot seat vibration begins at low amplitude, (beneath the normal background vibration level). The fore/aft hub forcing, airframe bending, pilot seat motion and AHRS motion form a closed loop feedback path that amplifies the 6 Hz vibration through biomechanically induced (6 Hz) motion of the collective stick and 6 Hz motion (vertical and lateral) of the AHRS. The amplification in pilot seat vibration and collective control oscillations was shown in **Figure 4**.

The complex nature of scissors mode amplification will not be discussed in detail. For the accident record, it is understood that the collective command, cyclic stick commands, and AHRS commands drove the amplification of the cabin vibration. In forward flight, oscillatory control inputs in these systems stimulate the scissors mode, which, although inertially reactionless, generates strong aerodynamic forces that are passed into the fixed system as fore/aft forces at the mast.

The significant out of plane blade motions in the accident record may be understood by considering the impact of the 6 Hz collective input on the main rotor. A collective control undergoing 6 Hz cycles, while turning a 4 Hz rotor, produces an anomalous control input. Collective (as the name implies) is intended to generate a uniform thrust increase on all blades and generally leads to simultaneous "coning" of the blade pattern in flight. In high speed forward flight, a high frequency oscillatory collective input (higher than the rotor's rotational frequency) will generate blade to blade thrust variations that depend on the timing of the blade's azimuthal position with the collective position. For example, if the stroking collective control is at its peak position when Blade No. 1 is on the advancing side of the rotor disk, then that blade will see a large lift increase. The very next blade (Blade No. 2) will arrive on the advancing position with the collective already into the down stroke and that blade see less lift. By the time Blade No. 1 returns to the advancing side (one revolution later), the collective will be at its minimum, and now Blade No. 1 sees a strong lift reduction (compared to the other blades). In response to the lift changes, out of plane blade flapping motion occurs. All five blades participate (to varying degrees) in the azimuthally varying lift arising from forward speed and the 6 Hz collective motion. Under these conditions, the familiar "rotor tip path plane" becomes blurred, with blades undergoing independent out of plane motion rather than smoothly tracking one another. The varying blade aerodynamic forces (due to the 6 Hz collective motion) also appear to further stimulate the scissors mode response of the main rotor, amplifying the airframe vibration and the collective biomechanical feedback response.

In addition to pilot biomechanical feedback, the airframe vibration also affected the Attitude Heading Reference Systems (AHRS) mounted in the forward cabin. In aircraft control systems, the general principal of using an AHRS is to detect uncommanded accelerations of the aircraft. For example, if a helicopter is hovering in gusting winds with the cyclic control held fixed, one might expect the helicopter to drift in response to the buffet forces. The AHRS senses the helicopter accelerations and makes a determination as to whether the pilot commanded the motion. Since the pilot's control did not move, the AHRS signal will be used to automatically supply appropriate control inputs that neutralize the uncommanded accelerations due to the gust. Since the AHRS operating bandwidth fell within the 6 Hz vibration frequency, the AHRS was able to participate in the airframe vibration response. Since the AHRS is mounted in the forward cabin, it sensed the combined "stirring" motion due to the 6 Hz vertical and lateral bending response of the airframe. In comparison to the sensed stirring motion in the forward cabin, the pilot cyclic control motions are small. Thus, the AHRS applied control commands to the main rotor. The 6 Hz stirring of the main rotor swashplate is another source of continued scissors mode stimulation.

3.1 Control Filtering

The AHRS units communicate with the control system to achieve numerous benefits, particularly in the arena of handling qualities and reduced pilot workload. Envelope expansion testing is generally focused on the nominal rotor rotational speed of 100% RPM. In the accident record, simulated single engine failure testing uncovered the vibratory feedback issue while remaining at ~92% rotor RPM. Although there were no structural or system failures prior to the tailboom strike, the feedback cycle that led to large amplitude cabin vibration was a root cause of the accident. The control laws are designed to account for known system modes and handle the biomechanical feedback scenario through control filtering methods. The control system inputs (Stick inputs and AHRS) are passed through filters that attenuate frequency content near the system modes. Setting the filter depth and frequency band involves a careful exercise in attenuating the undesirable frequency content without negatively impacting the aircraft handling qualities. For operation at 92% RPM at high forward speed, the filters did not sufficiently attenuate the 6 Hz control system inputs passed to the main rotor actuators.

For the cyclic control, notch filters were placed near the airframe structural resonance frequencies to attenuate the ability to pass undesirable excitation frequencies to the rotor swashplate actuators while permitting low frequency control inputs to pass through and control the aircraft. In the pilot's collective axis, no filtering was present in the prototype aircraft at the time of the accident. In general, filters are added precisely and only as necessary, both to avoid unnecessary complexity and to avoid negatively impacting the aircraft handling qualities. This approach, established over many years of development activity at Bell Helicopter had demonstrated no previous need for a collective filter. Without a filter in the test aircraft, all 6 Hz collective control motion (except for instances of rate limiting discussed next) was passed to the main rotor actuator. It is possible that biomechanical feedback could be attenuated by releasing the collective control or by higher friction settings on the control stick. However, there was no indication that the pilot attempted to release the control at any time.

There were several instances in the accident record where the pilot collective oscillation was attenuated by collective rate limiting. At times where the vibration was the highest (11.5 seconds and 17.5 seconds in **Figure 4**), the biomechanical collective motion produced collective rates that exceeded a prescribed limit in the control laws. Figure 11 shows the effect of collective rate limiting. The green curve represents the rate-limited collective control that was applied to the main rotor actuators during the accident record. Without rate limiting, the larger cycles of the blue curve would have been sent to the main rotor actuator. By comparing Figure 11 with Figure

4 (Pilot seat vibration), it is seen that when collective rate limiting was in effect, the diverging vibration blooms reverted to lower vibration levels. The rate limiting appears to damp the feedback cycle, reducing the vibration level from 2.5*g* to 1*g* shortly after 11.5 seconds and again at 17.5 seconds. It is believed that since the damping of vibration coincided with the forward cyclic and positive collective at 11.5 seconds, the pilot may have associated the vibration reduction with those control inputs. This "negative training" may explain the second application of forward cyclic control and positive collective during the second vibration bloom at 17 seconds (assuming the pilot was attempting to reduce vibration levels).

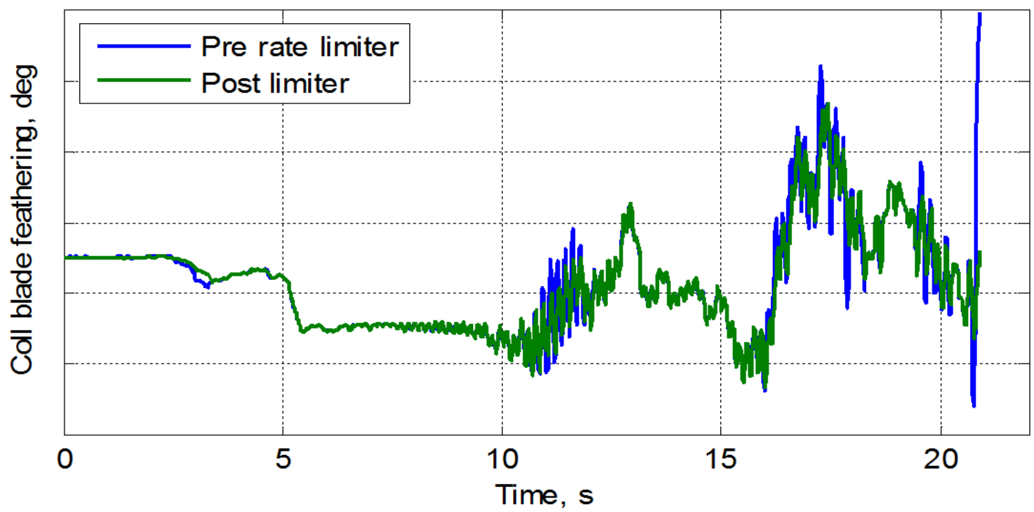

Figure 11. Biomechanical feedback of the collective control is inhibited by collective rate limiting enforced by the control laws. Rate limiting reduced the oscillatory collective input applied at the main rotor actuators and appeared to dampen the feedback cycle, reducing pilot seat vibration.

4. CONCLUSIONS

While at a high airspeed test point, a sustained low rotor RPM allowed for the development of high vibration throughout the aircraft. The high vibratory environment led to adverse control inputs that reduced the rotor RPM to critical levels and resulted in high flapping of the rotor blades. A near full aft cyclic control input led to one of the rotor blades severing the tail boom and loss of control of the aircraft.

In the accident record, components of the helicopter interacted as a closed-loop dynamic system. A main rotor scissors mode transmitted vibrations into the airframe through the rotor mast. The vibratory forces were transmitted throughout the airframe, including the pilot seat and forward cabin, at 6 Hz. At the pilot seat, these vibrations were passed to the control sticks through an unintentional biomechanical response. The vibratory motion in the forward cabin was sensed by the Attitude Heading Reference System (AHRS), and a corresponding 6 Hz signal was passed to the flight control system, which responded with a 6 Hz command into the main rotor swashplate. The 6 Hz biomechanical and AHRS-induced rotor control commands served as unanticipated excitation amplifiers of the main rotor scissors mode, closing the feedback path on the system.

5. GOING FORWARD

As the 525 flight test program continues its development, the design will incorporate a new collective control filter that targets vibration frequencies near the 6 Hz band. Existing filters in the cyclic control are tuned and deepened, as are the rate filters in the AHRS system. Enhanced closed loop system modeling shows that these filter changes would result in a damped response that attenuates control system feedback and stabilizes cabin vibration throughout the flight envelope (and outside the flight envelope where the steady 92% RPM led to the amplification of the main rotor scissors mode). Evaluations of the new filters in the 525 flight simulator indicate that the impact of the filters is detectable from a handling qualities perspective, but not objectionable.

Significantly improved analytical capability has been developed for the telemetry monitoring stations. Going forward, a live-streamed decomposition of all active rotor modes will be presented on the display monitors. When different rotor modes become simultaneously active during a test, the flight test team will have immediate assessments of rotor response and specific modal damping. An incremental test approach is planned for evaluating rotor damping characteristics at both 100% RPM and reduced steady state RPM. The testing will utilize a new engine trim file that permits rotor RPM to be adjusted as low as 90% RPM. With this engine trim file, full engine power will be available at reduced RPM for evaluating rotor damping without the need for simulated single engine failure scenarios. Simulated single engine failure testing, throughout the flight envelope, will still be completed separately per existing test plans.

To support future simulated single engine failure testing during the remaining development testing, the Special OEI Training Mode of the test aircraft will incorporate a “kick-out” that automatically exits the training mode if the rotor RPM ever drops below 80%. This feature would restore full (dual) engine power and recover rotor RPM to 100% without pilot intervention. The 525 Production OEI Training Mode always planned a “kick-out” near 90% RPM. Both the test aircraft and the 525 production aircraft will also feature a button on the control stick to facilitate disengaging the OEI training mode.

For improved situational awareness, particularly in a high vibratory environment, a new high visibility dedicated “Low Rotor RPM” warning light is added to glare shield of the test aircraft and will become the 525 production standard. Previously the Low RPM warning was part of the pilot’s primary display.

For the accident record, controls were flown without stick trim forces (pilot pressed the force trim release buttons on each stick). The reasoning behind the use of “Force Trim Release” in the cyclic control is as follows: When the aircraft is trimmed at the V_H speed, a dive must be initiated to achieve the V_{NE} speed (a difference of ~20 knots for this test). The test aircraft control system had a maximum “speed hold” limit near V_H and so, any speed over that value required the pilot to hold the higher speed. The pilot could choose to push against the cyclic forces for the entirety of the test point. Alternatively, the pilot could press the stick force trim release (FTR) button to remove the forces and eliminate the back drive, permitting more precise targeting of the desired airspeed.

In the collective axis, the control system applies a tactile cue any time the rotor droops below 103% RPM while on the 30 second engine limit. The collective stick tactile cueing, has two effects. The tactile cueing increases the friction force by a fixed amount and also attempts to drive the stick down (so that power available is equal to the 30 second engine limit). Once the power required is below the engine limit the "additional" friction force is removed and the stick stops driving down. At the time of the accident, releasing the collective "trim force" had the effect of removing the "back driving," but did not remove the additional friction force. Pressing the collective FTR button, as done in the accident record, permitted more precise control of power to set the test point initial conditions. The effect of flying the test point with FTR took away the backdrive cueing that would encourage the pilot to lower the collective to eliminate the rotor droop. To improve situational awareness in the future, tactile backdrive cueing will remain active when the FTR button is pressed.